BETTER, FASTER, CHEAPER PLANETARY MISSIONS USING SOLAR ELECTRIC PROPULSION

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Abstract

Planetary missions of the future will have increasingly greater energy requirements due to the desire for *in-situ* investigations and faster flight times. Solar electric propulsion provides a means of more effectively accomplishing these types of missions. The first planetary low-thrust mission will fly by an asteroid and a comet, and will be launched in 1998. This mission will demonstrate solar electric propulsion technology, and lay the groundwork for more exciting missions in the future. Three missions representative of the types of future missions for which solar electric propulsion might be used are shown. The performance of solar electric propulsion systems are compared to chemical systems, with the solar electric significantly outperforming the chemical system in each case.

Introduction

The strategy of how to explore the solar system is currently undergoing revolutionary changes. Previous missions have flown by all the planets except Pluto, and have also visited several comets and asteroids. These initial reconnaissance missions have returned a wealth of scientific information, but the next stage of' exploration will require *in-situ* investigations with planetary orbiters, surface rovers, and sample returns. This phase of planetary exploration began with missions like Viking, Galileo, and Cassini, and are considerably more challenging from an engineering standpoint than simple flyby missions. There is an obvious desire for larger spacecraft with more instruments to probe every facet of a remote surface. Also, and more directly related to the subject of this paper,

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the propulsive requirements to perform these missions are considerably more demanding. This may require a rendezvous and planetary capture or a fast flyby (or impactor) of some object in the outer solar system (i.e. Pluto).

On the other hand, the need to perform good plane.tary science on smaller budgets is driving, the industry to smaller launch vehicles, less complicated flight systems, and shorter missions. 1 low can the realities of smaller budgets be reconciled with the. scientific requirements for more challenging missions? We suggest that part of the answer is more efficient propulsion systems. Current solar electric systems can deliver specific impulses on the order of 3000 sec, as opposed to around 310 sec for space storable chemical propellants. Of course one dots not get a ten-fold increase in performance, because the. long thrust arcs of a solar electric propulsion (S1 i]') system suffer more from gravity losses. Nevertheless, this difference is too large to ignore for missions with significant post-launch propulsive requirements. The New Millennium Deep Space 1 (DS1) mission [1] will fly by an asteroid and a comet, will demonstrate the viability of solar electric propulsion technology, and will pave the way for some truly exciting missions in the next several decades. in the following, we will compare the performance of conventional (or chemical) trajectories with SEP trajectories. The weds chemical and conventional will be used interchangeably.

The acquisition of launch services is one of the largest cost elements for an interplanetary mission. There is always much interest in flying on the smallest (least expensive) vehicle. Below we will show examples of three missions of current interest which if done conventionally would require at least a current Delta II 7925 capability: a rendezvous of the main belt asteroid Vesta, a Pluto flyby, and a Phobos sample return. We show (hat several of these missions could be performed on the much smaller (- half the performance) and less expensive Delta II 7326, even assuming a small flight system similar to that planned for DS 1. The cost of the Delta 7326 is currently estimated in the range of

$40M to $45M, while the Delta 7925 is between $50M and $55M. Also, in cases where the launch vehicles are the. same, the delivery capability of the SEP mission is vastly superior.

Optimizing continuous thrust trajc.dories is more difficult than that of optimizing impulsive trajectories in that the thrust vector is continuously changing over large. portions of the mission as opposed to optimizing a relatively small number of "discrete" impulses. At JPL, the tool used to optimize these trajectories is VARITOP [2 ,3]. It uses a traditional "calculus-of-vari ations" (COV) approach and has been a "workhorse" for solving this problem for many years. Planetary flybys are capable of significantly leveraging mission performance for either chemical or SEP trajectories. Unfortunately, experience has shown that the convergence of the COV approach is extremely sensitive to numerical perturbations which arise when planetary flybys arc added to the problem. Therefore, any technique which dots not have this sensitivity and either replaces the variational technique or provides some knowledge of the optimal solution from which to start the variational approach would be very beneficial. Wc have reported in the past a new approach for optimizing these trajectories [4]. Wc used this technique to provide a starling point for several of the trajectories below which were. subsequently optimized with the. variational code.

In this paper wc will compare the performance of SEP and chemical flight systems. By performance, wc mean either the burnout or the propellant mass required to deliver it to the. required destination. The sum of these two is the injected mass determined from launch vehicle *performance* curves. For chemical systems the rocket equation is applied to a separately optimized trajectory to determine the burnout and propellant masses. For the SEP systems, a mass optimization is performed by VARITOP such that the burnout mass is maximized by varying the thrust profile and planetary encounter conditions. The launch C_3 (and therefore injected mass) may also be varied by the program in order maximize burnout mass. We make no attempt in this paper to quantify either the mass or cost of various subsystems. These estimates will come from a more detailed analysis of each mission, and will be dependent on the level of technology assumed in that analysis. We chose to preserve the generality of our results, rather than attempt to make these estimates.

First SEP Mission

The first interplanetary mission to use SEP for its main propulsion will be DS1. This will be the first in a series of technology validation missions in the New Millennium Program. Currently, DS1 is scheduled to launch in July 1998. It will fly by both an asteroid (3352 McAuliffe) and a comet (West-Kohoutek-Ikemura (W-K-I)) over a two year period. The trajectory is shown in Figure 1. The dotted lines in the figure represent orbits of the bodies involved (Earth, McAuliffe, and W-K-I). The solid lines along the trajectory are thrust arcs, and the spacecraft is coasting along the dashed arcs. Tic marks along the trajectory arc shown at 30 day intervals. *Icons* along the trajectory represent discontinuities, which may be planetary flybys, thrust/coast transitions, or thruster throttling[3]. This mission, which will be launched on a Delta 7326, has ample mass margin. The surplus launch capacity will probably be used for two secondary payloads. At the time this paper is being written, the current estimate of spacecraft mass is about 388 kg.

FIG. 1- First Interplanetary Solar Electric Mission (DS 1),

The primary purpose of this mission is to validate several key technologies, as discussed in Reference [1]. The NASA SEP Technology Application Readiness (NSTAR) program is a joint JPL/Lewis Research Center effort to validate low-power ion propulsion

[3] The ion propulsion system (IPS) adjusts voltage, beam current, and mass flow rate to achieve a desired thrust within 16 discrete intervals (or states) between a maximum and minimum input power. As the power varies with solar range, the IPS "throttles" between successive states.

which was begun in 1992'. The NSTAR system will accelerate Xe^+ ions through a molybdenum grid in a 30-cm thruster, producing about 90 mN Of thrust at maximum thruster power (2.3 kW) and about 20 mN at minimum power (0.52 kW). Thruster input power is reduced from that provided by the solar array due to inefficiencies in the power processing unit. The efficiency of this system (including the power processing hardware) varies considerably with power available. It achieves a specific impulse of' about 3280 sec at maximum power, and about 1990 at minimum power. Power will be provided by another new technology item, the Solar Concentrator Arrays with Refractive Linear Element Technology (SCARLET), which is being provided by the Ballistic Missile Defense Organization. This array uses cylindrical Fresnel lenses to concentrate sunlight onto $GaInP_2$/GaAs/Ge cells arranged in strips expected to achieve at least 24% efficiency. Throughout the paper, unless otherwise specified, values given for power available are values normalized to 1 astronomical unit (AU).

SEP typically outperforms conventional propulsion systems for missions with larger propulsive requirements. The energy requirements of this mission are rather moderate., so that SEP is not an enabling technology, but it is expected to be extensively exercised on DS 1. It is hoped that a successful demonstration of SEP on DS 1 will lead to the use of this technology on much more challenging missions like those discussed below.

Future Missions

Vesta Rendezvous

An excellent example of the strong desire for in-situ investigations mentioned above is a main belt asteroid rendezvous. The Galileo mission to Jupiter flew by Ida and Gaspra. NEAR will fly by Mathilde, and rendezvous with the near-Earth asteroid Eros. Among other things, investigators want to measure gross properties such as mass, volume, and density, characterize the object's surface, and perform high resolution multi-spectral mapping of the surface [5]. These activities require a closed orbit and weeks to months of investigation. Vesta is one of the most attractive targets for such a mission. It is one of the largest (main belt) asteroids (?55 km radius), and is believed to be the parent body for a certain class of mete.oritcs (eucrites). Evidence which links meteorite samples to known asteroids is available in only a few cases. It would be highly desirable to closely examine an asteroid surface which we suspect to be the pare.nt hod y of meteorite samples.

Using chemical propulsion}, this mission is too demanding for the Discovery class missions being studied today. '1'able 1 shows a performance comparison of three conventional trajectories with two SEP trajectories. Most of the opportunit ics shown launch in 2005 because of the favorable Mars-Vesta geometry in that year. An extensive study of double-Mars gravity-assist chemical trajectories to main belt asteroids has been conducted [6]. '1'here are no double Mars opportunities to Vesta in Ref. 6 which launch in 2005, so a 2003 opportunity is strewn. Double Mars fl yby trajectories characteristicall y have long flight times, but offer the best possible chemical performance to main belt objects. ~'able 1 shows performance on the Delta 7326 and Delta 7925. The direct and single Mars gravity-assist conventional trajectories have inadequate performance to be of any practical consideration, even on the larger Delta 7925. It may be possible to fly this mission on the large.r 7925 using a double Mars option, but this would require a very long flight time (6.3 yrs). The two SEP options have good mass performance combined with short flight times, and therefore clearly offer a superior mission.

Figure 2 shows the performance of the two SEP trajectories as a function of flight time on each launch vehicle. The burnout mass includes all mass except the xenon propellant. The fast flight times around 1.3 years involve a range. of C_3 around 4 to 6 km^2/s^2 cm the Delta 7925 increasing up to 55 to 65 km^2/s^2 near 2.8 years. The range of ('3 on the Delta 7326 is smaller (0,7 to 30 km^2/s^2). The performance curves begin changing slope at flight times bc.[wee.n 1.7 and 2.() years. Where the curves are flat (Delta 7326) nothing is really gained by the. longer flight time. The rendezvous reall y occurs around 1. 8

years, and for longer flight times, the remainder is just a coast phase. In the case of the Delta 792S curves, the change in slope corresponds to the introduction of an additional coast arc. In each case, the Mars gravity-assist trajectory (solid lines) delivers more mass while using less propellant (except for very fast flight times where the performance is rather poor anyway).

TABLE 1 - SEP vs. Chemical Performance for Vesta Rendezvous

TRAJ	PROPULSION	LAUNCH DATE	FLIGHT TIME (YR)	DELTA 7326 BO / PROP* (KG)	1)1ELTA 7925 BO / PROP* (K(i)
1DIRECT	CHEMICAL	JUL 2005	4.1	64/159	1421354
MARS	CHEMICAL	AUG 2005	1.8	1021276	208 / 561
MARS-MARS	CHEMICAL	JUN 2003	6.3	150/216	307 / 442
DIRECT	SEP	SUMMER 2005	2.0	405 / 146	596/171
MARS	SEP	SUMMER 2005	2.0	436/113	6(E3 / 123

4: BO / PROP - burnout mass / propellant mass (sum of these numbers is injected mass)

FIG. 2- SEP Performance for a Vesta Rendezvous Trajectory.

Figure 3 shows the direct SEP trajectory launched on the Delta 7326 shown in Table 1. It has a two year flight time with ii low launch energy ($C_3 = 0.93$ km^2/s^2). This trajectory optimizes to one with two coast arcs between the three "continuous" thrusting segments. Note that the launch C_3 is an optimized parameter. VARITOP maximizes the burnout mass by modifying the thrust profile (direction of the thrust vector as well as the duration of the coast/thrust segments), and optimizing C_3 subject to the constraint of the launch vehicle performance. "1 he Mars gravity-assist version is shown in Figure 4. It has a C_3 of 1.01 km^2/s^2, and also has a 2 year flight time. Comparing Figures 3 and 4, one can see that the Mars flyby "replaces" the second thrust arc on the direct trajectory, thereby accounting for the reduction in propellant consumption mentioned above.

FIG. 3 - Direct SEP Vesta Rendezvous Trajectory (Delta 7326).

FIG. 4- Ma Is Gravit - Assist Trajectory to Vesta (Delta '/3?,6).

Pluto Flyby

In addition to rendezvous missions, it is also possible to use SEP to increase the heliocentric energy of a trajectory to achieve fast flight times to the outer solar system. One application of current interest would be the Pluto Express mission. Considerable effort has been devoted to finding good trajectory oppor tunities for this mission [7]. The project is currently planning to use a conventional chemical propulsion system. The primary mission launches in March 2001, with a backup mission opportunity in July 2002. The backup is a product of Reference [7]. Each has a 12 year flight time. The primary trajectory includes three Venus flybys prior to the Jupiter encounter, and the backup only two, though the performance is less on the. backup. The key to these. trajectories is a Jupiter encounter which provides the benefit of a large gravity-assist. The set of opportunities is restricted to only a few years due to the geometry between the outbound trajectory, Jupiter, and Pluto, along with the need to keep the Jupiter flyby radius beyond about six Jupiter radii (R_j) to minimize the exposure of spacecraft electronics to Jovian radiation. For reasonable flight times, these trajectories are generally characterized by large deterministic maneuvers (- 2 km/s) near the final Venus flyby. Large maneuvers that near the sun are undesireable because they may impose additional requirements (it: thermal, pointing, etc . ..) on the space.cra.fl design. A spacecraft mass of about 360 kg can be delivered on a Delta 7925 with the.plimary mission, but only about 260 kg is possible with the backup.

An excellent SEP alternative for this mission, which launches in July 2002 on a smaller Delta-Lite (a vehicle intended to be smaller and less expensive than the Delta 7326), has been found by other investigators at 1])[1], [8]. It delivers 370 kg (burnout mass) with a 3.3'/5 kW solar array. It is the S10' version of the current chemical backup trajectory, but has a faster flight time, and significantly better performance: (370 kg in 10.2 years on a small launch vehicle vs. 260 kg in 1 ? years on a Delta 7925).

One of the criticisms of SEP for this mission was that no good backups were known. We looked for simpler trajectories with one or no Venus flybys, but permitted more complicated flight systems involving two or three simultaneous thrusters, and larger solar arrays. We found direct trajectories (no Venus flybys) in 2003, and trajectories with one Venus flyby launching in 2004.

Figures 5 and 6 show the results for trajectories which launch in 2003, and go directly from Earth to Jupiter. In Figure 5, two configurations are considered: 1) a 6 kW array using 2 thrusters launched on a Delta 7925, and 2.) an 8 kW array with three thrusters launched on a Delta 7326. As before, the burnout mass includes everything except the Xe propellant, and is shown as a function of flight time to Pluto. The Jupiter flyby radius and Xe propellant requirement are shown in parentheses. The smaller launch vehicle is probably slightly short on performance, even with the larger solar array and additional thruster. Nevertheless, there are some nice mission options here on the Delta 7925 down to flight times of under 10 years. Figure 6 shows the sensitivity of this data to solar array power, Three solutions from the upper curve in Figure 5 were computed with a range of solar array powers. The solid vertical line in the middle intersects the curves at 6 kW - which reflects the value in Figure 5. An example of one of these trajectories is shown in Figure ?.

FIG. 5 - SEP Performance : Jupiter Gravity-Assist to Pluto vs. Flight Time.

FIG. 6 - SEP Performance : Jupiter Gravit - Assist to Pluto vs. Solar Array Power

FIG. 7- SEP Jupiter Gravity-Assist Trajectory to Pluto.

Figures 8 and 9 show the results for tmjc.clerics in 2004. These include a Venus gravity-assist prior to going to Jupiter. The two launch vehicles previously considered are examined here, with both the 2 and 3 thruster configurations on the spacecraft. Figure 8 shows sensitivity to flight time, and figure 9 displays sensitivity to solar array power. All trajectories displayed in Figure 8 have an array power of 6 kW. 'The values in parentheses are Jupiter flyby radius ant] Xe propellant as before. In Figure 9, the effect of solar array power is shown by varying the power for each solution at the endpoints Of the curves in Figure 8 - thereby bounding the problem. Pluto flight times arc shown in pare.l~thcscs.

With the Venus gravity- assist, the Delta 7326 with 2 thrusters is still a bit shy on performance, though the larger arrays help somewhat (378 kg at 8 kW vs. 361 kg at 6 kW for a 12 year flight time). For equal flight times, a third thruster gives approximately a 50 kg performance advantage., This performance benefit will be intriguing if the mass penalty of the additional thruster and associated hardware. is not too extreme. The significant "leveraging" ability of the SEP is accentuated by the fide.t that one gets more performance. advantage by adding a third thruster, than by going to a much bigger launch vehicle (see the. two middle curves in Figure 8). It would be most intriguing if a way could be found to fly this mission on the smaller Delta 7326. This is why the 8 kW configuration is shown in Figure 5, even though all other curves in Figures 5 and 8 are at 6 kW. Even at this higher power level, the performance is probably too low.

It is also interesting that there is no particular advantage to the Venus gravity-assist for the 2 thruster configuration. This is seem by comparing the 2 thruster / Delta 7925 curves in Figures 5 and 8. Though these are in different years, the performance of direct trajectories in 2003 and 2004 are very similar (direct trajectories in 2004 were computed, but are not shown). The Venus gravity-assist opportunities shown in Figure 8 are not available in 2004." A 6 kW / 3 thruster / Delta 7925 configuration was also considered for the non-Venus trajectories. It is not shown in Figure 5 because it was only slightly better than the 2 thruster curve shown. This is because the 6 kW array does not provide enough power to adequately drive the three thrusters. Larger solar arrays would fare better. Therefore, unlike the 2 thruster case, for the 3 thruster / 6 kW configuration on the Delta 7925, the Venus gravity-assist dots significantly enhance the performance. The advantage of both a larger launch vehicle and an additional thruster is quite significant, delivering some impressive payloads in short flight times. An example of one of these trajectories is shown in Figure 10.

FIG. 8 - SEP Performance : Venus-Jupiter Gravity-Assist to Pluto vs. Flight Time.

FIG. 9- SEP Performance : Venus-Jupiter Gravity- Assist to Pluto vs. Solar Array Power.

FIG. 10- SEP Venus-Jupiter Gravity-Assist Trajectory to Pluto.

Phobos Sample Return

Finally, we will consider a SEP sample return mission to the Martian moon Phobos. As already stated there is much interest in studying primitive, undifferentiated solar system main belt asteroids. The Martian moons, Phobos and Deimos, may offer an attractive alternative to a main belt mission. They are believed by many to be captured C-type asteroids, Also, Mars is much closer to the sun, where it is much easier to operate the SEP.

We compare again a chemical with a SEP option. Figure 11 shows a conventional trajectory which launches in October 2000 with a C_3 of 10.35 km^2/s^2. The Earth-Mars leg is a long flight time Type IV (transfer angle between 1.5 and 2.0 heliocentric revs) which arrives at Mars on 21 Feb 2003. It was chosen because it has a low Mars arrival velocity (V_∞ = 2.5 km/s). Other opportunities in this time period have lower launch energies and shorter flight times, but result in a Mars arrival V_∞ around 4.6 km/s and greater. The optimum Earth return opportunity is on 18 April 2003, so that 56 days are available for rendezvousing with Phobos, collecting a sample, and preparing for departure. A Type 1 return leg arrives at Earth on 10 Nov 2003 with an arrival V_∞ of 3.0 km/s, slightly more than three years after launch.

FIG. 11 - Conic Phobos Sample Return Trajectory.

The simplest strategy for Mars orbit insertion is to directly insert into orbit with Phobos. This requires an insertion ΔV of 1.76 km/s. Certainly, other strategies might be considered such as a series of chemical maneuvers combined with aerobraking passes through the. Martian atmosphere, or in the limit, one aerocapture pass with a chemical maneuver to raise periapsis and rendezvous with Phobos. Table 2 compares the SEP option with two chemical options. The most conservative is all chemical with no aerobraking and as one would expect yields the poorest performance. in terms of performance, the best possible chemical scenario would use an aerocapture pass to remove enough energy from the trajectory such that it leaves the Mars atmosphere on a trajectory with an apoapsis at Phobos, needing one chemical maneuver to rendezvous. This gives much better mass performance than the other chemical option, but still considerably less than the SEP. As was done in the Vesta example above, results are shown for a Delta 7925 and a Delta 7326.

TABLE 2 - SEP vs. Chemical Performance for Mars Sample Return.

STRATEGY	LAUNCH DATE	TIME AT PHOBOS (DAYS)	MISSION DURATION (YR)	DELTA 7326 BO / PROP* (KG)	DELTA 7925 BO / PROP* (KG)
CHEMICAL	OCT 2000	56	3.1	140 / 328	279 / 656
CHEM WITH AEROCAPTURE @ MARS	OCT 2000	56	3.1	195 / 273	390 / 545
SEP**	NOV 2000	56	3.0	392 / 181	798 / 349

* BO / PROP - burnout mass / propellant mass (sum of these numbers is injected mass)
** Delta 7326: 4 kW array, 1 thruster; Delta 7925: 8 kW array, 2 thrusters

The SEP trajectory is shown in Figure 12. It launches on 26 Nov 2000 with a C_3 of 1.36 km^2/s^2. It coasts for two months before beginning a long thrust phase. The spacecraft rendezvous with Mars in Feb 2002, spirals in to Phobos over 133 days, spends 56 days at Phobos, and spirals out over 158 days. The model used to estimate planetary spiral trajectories in VARITOP is described in Reference [9]. The departure spiral takes longer because Mars is moving away from the sun so that less solar array power is available. The mission ends three years after launch on 26 Nov 2003, and arrives back at Earth with a V_∞ of 2.6 km/s. The three year flight time was arbitrarily chosen (flight time not optimized), but the launch and arrival dates are optimum subject to this three year flight time. The flight system requires a 4 kW solar array and only one thruster operating at a time. The 56 day Phobos stay time was chosen for the SEP for purposes of comparison with the chemical mission. Actually, both the SEP and chemical trajectories are rather insensitive to changes in stay time over a range of 30 to 90 days. In each case, the burnout mass varies 7 to 8 kg over this range.

FIG. 12 - SEP Phobos Sample Return Trajectory.

conclusions

This paper takes a current look at several challenging missions of high scientific interest, and compares the. performance requirements imposed on the mission for both a chemical and a solar electric main propulsion stage. We consider a Pluto flyby mission, a Phobos sample return, and a rendezvous with the main belt asteroid Vesta. We show that the increased performance offered by solar electric propulsion enables shorter flight times, fewer planetary gravity-assists, or smaller launch vehicles, and in some cases - all the above.

We emphasize that in the future, faced with the requirements of increasingly demanding missions in an environment of shrinking budgets, mission planners will increasingly turn to propulsion systems such as SE}' which offer these significant performance advantages. The first use of solar electric propulsion as a main propulsion stage for a planetary mission will be on the New Millennium Program's Deep Space 1 mission in 1998. Deep Space 1 is designed to validate this technology (as well as several others), and to "open the door" for using it on missions such as those described in this paper.

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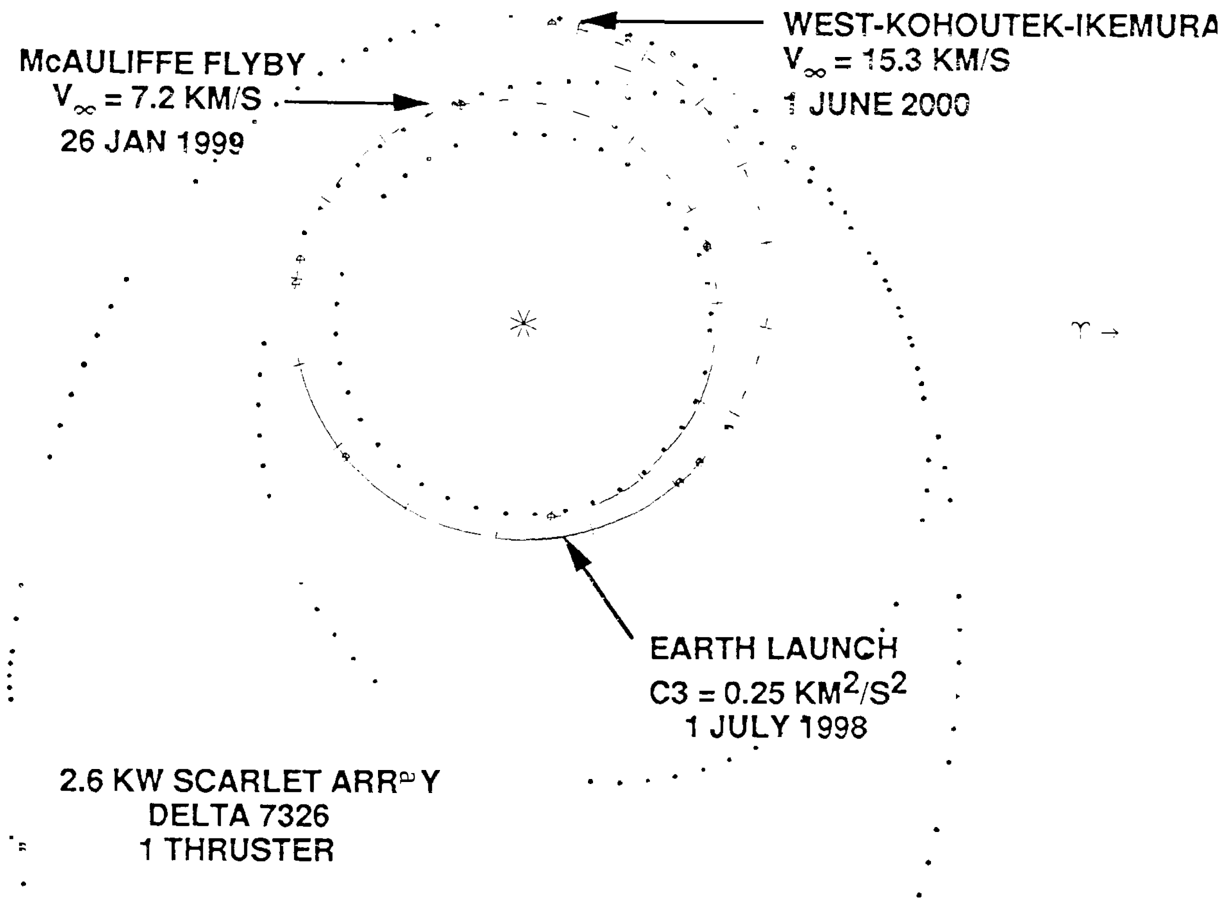
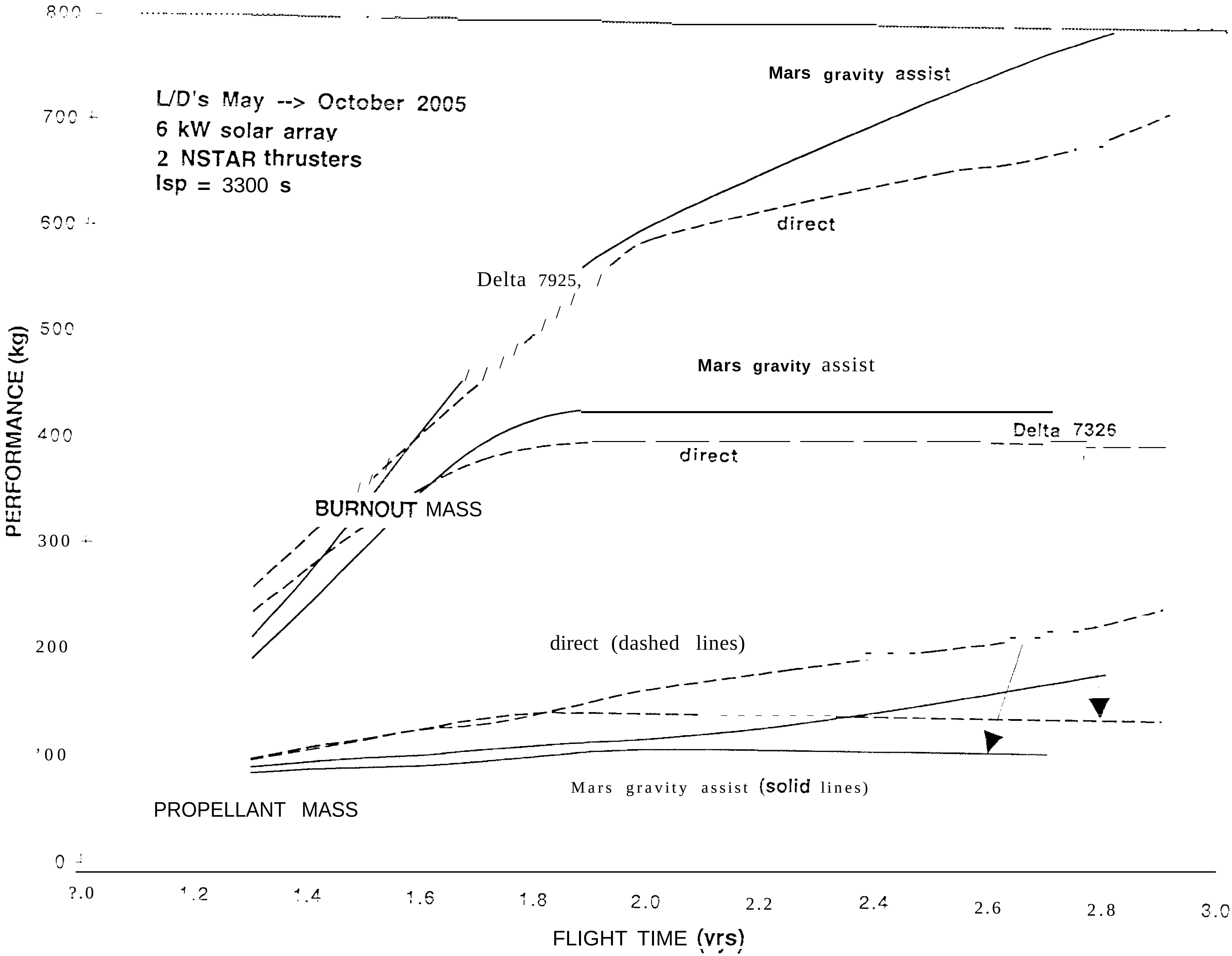
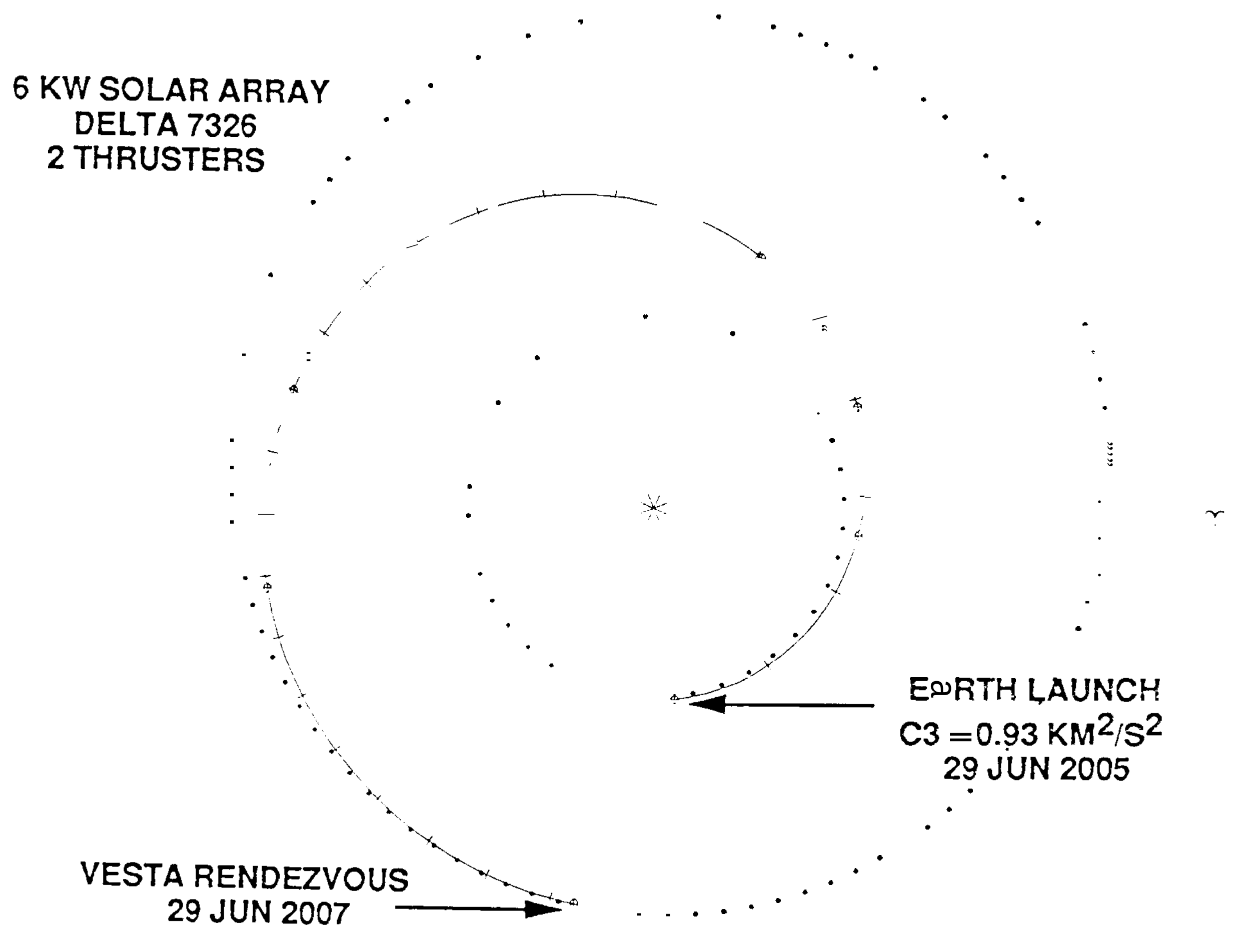

McAULIFFE FLYBY
V_∞ = 7.2 KM/S
26 JAN 1999
WEST-KOHOUTEK-IKEMURA
V_∞ = 15.3 KM/S
1 JUNE 2000
EARTH LAUNCH
C3 = 0.25 KM^2/S^2
1 JULY 1998
2.6 KW SCARLET ARRAY
DELTA 7326
1 THRUSTER

Fig 1



L/D's May --> October 2005
6 kW solar array
2 NSTAR thrusters
Isp = 3300 s
Mars gravity assist
direct
Delta 7925,
Mars gravity assist
Delta 7326
direct
BURNOUT MASS
direct (dashed lines)
Mars gravity assist (solid lines)
PROPELLANT MASS
PERFORMANCE (kg)
FLIGHT TIME (yrs)
800
700
600
500
400
300
200
'00
0
?.0
1.2
1.4
1.6
1.8
2.0
2.2
2.4
2.6
2.8
3.0

Fig 2


6 KW SOLAR ARRAY
DELTA 7326
2 THRUSTERS
EARTH LAUNCH
C3 =0.93 KM2/S2
29 JUN 2005
VESTA RENDEZVOUS
29 JUN 2007

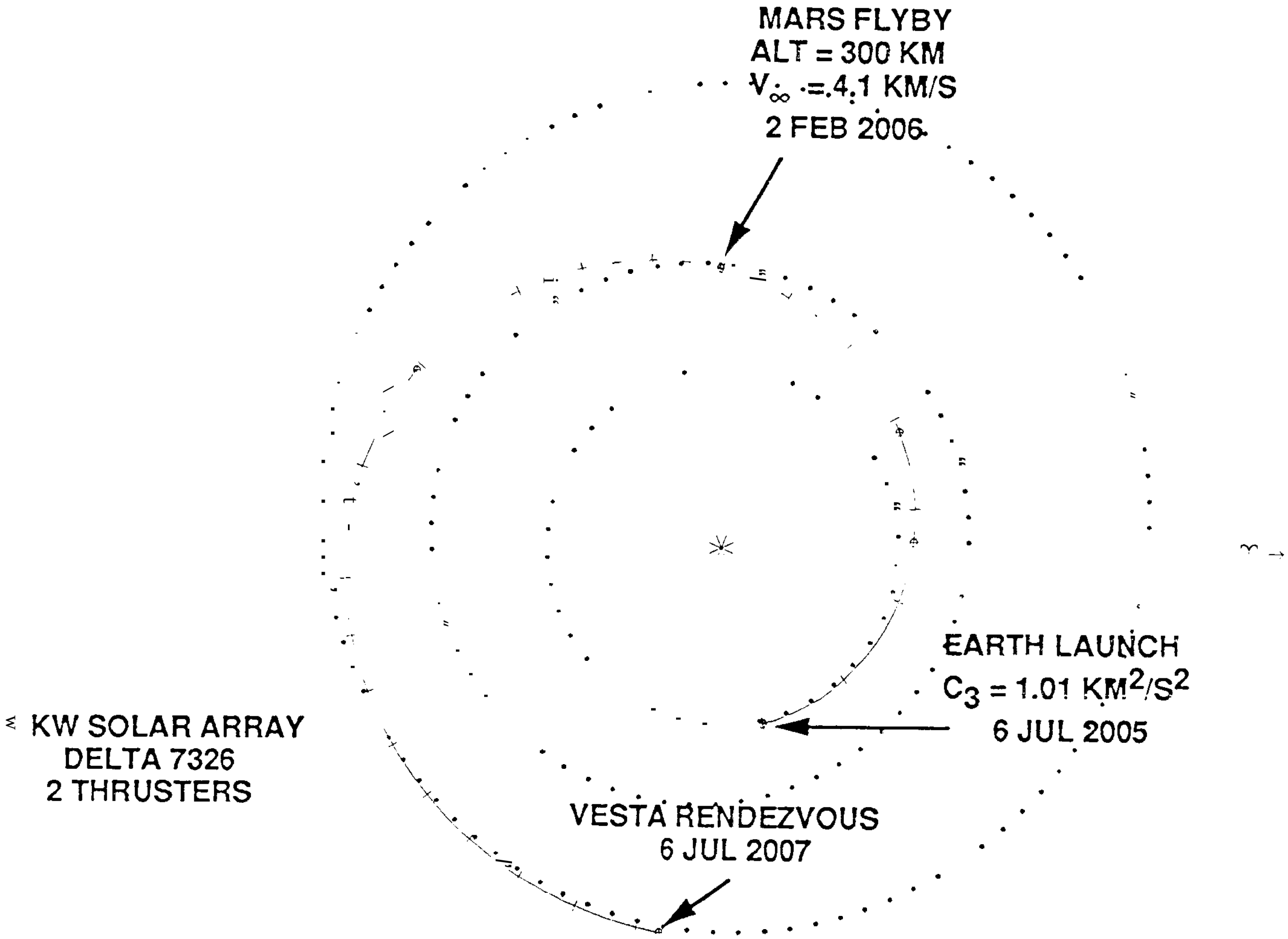

MARS FLYBY
ALT = 300 KM
V_∞ = 4.1 KM/S
2 FEB 2006
EARTH LAUNCH
C_3 = 1.01 KM^2/S^2
6 JUL 2005
VESTA RENDEZVOUS
6 JUL 2007
KW SOLAR ARRAY
DELTA 7326
2 THRUSTERS

Fig 4

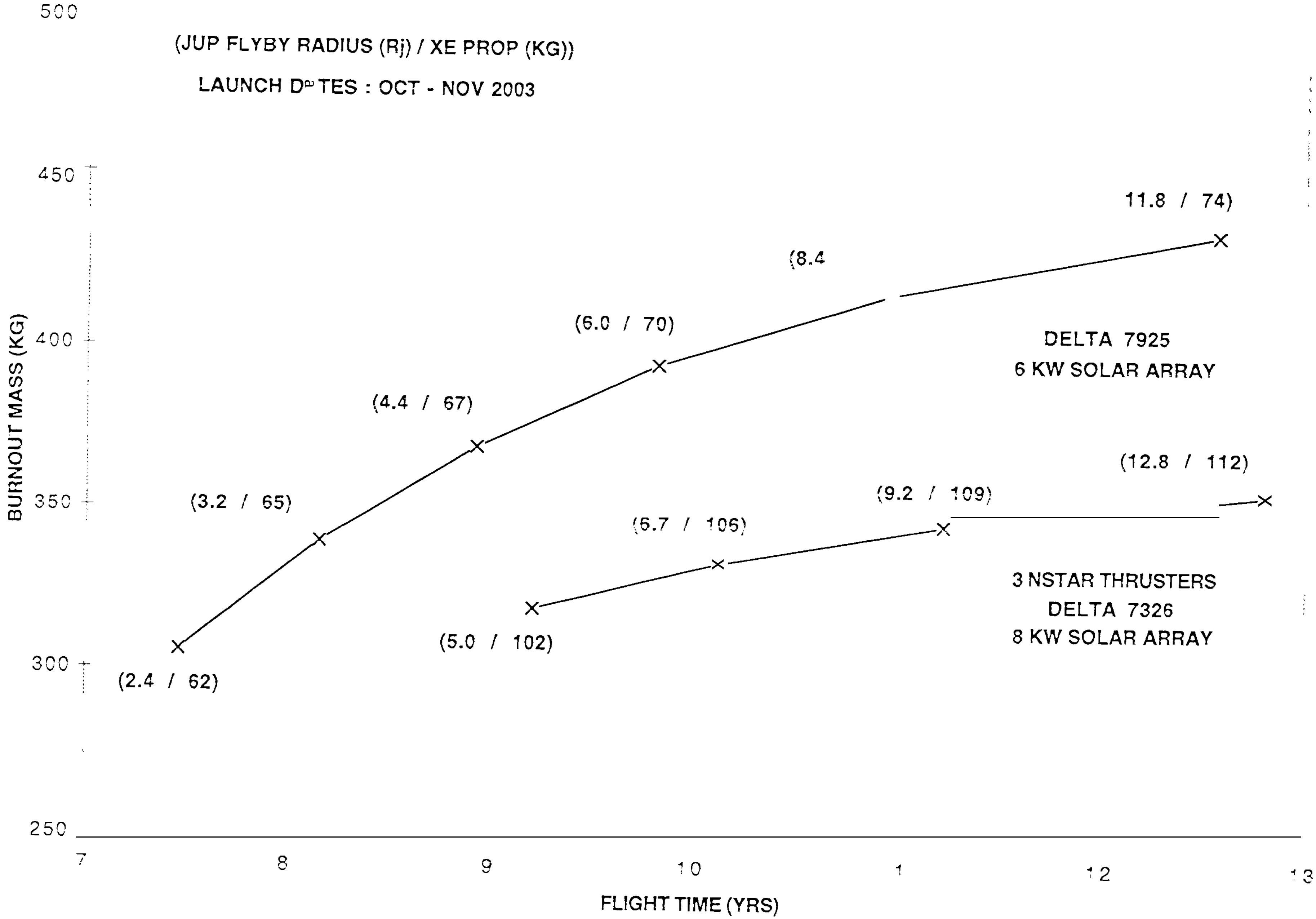
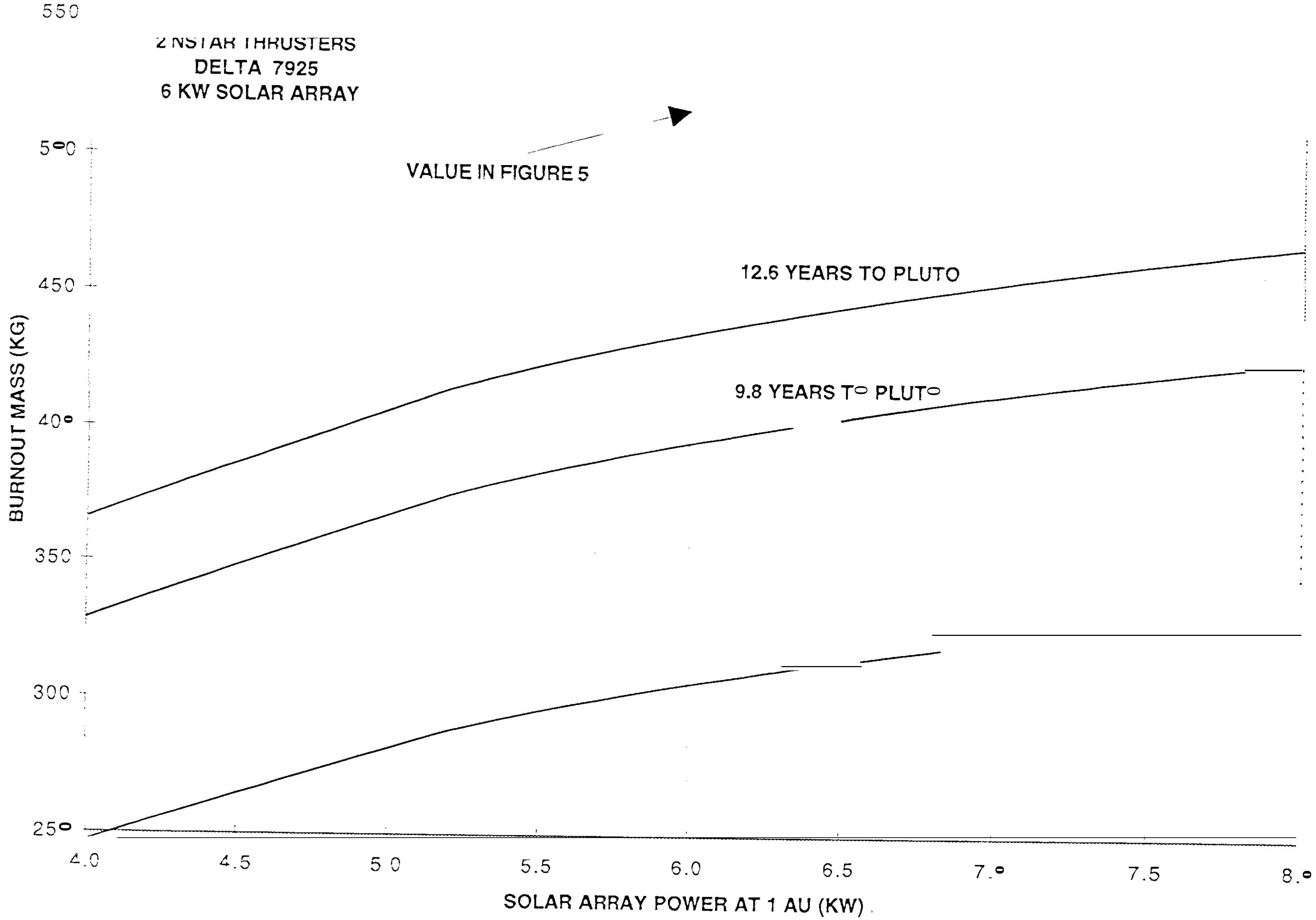
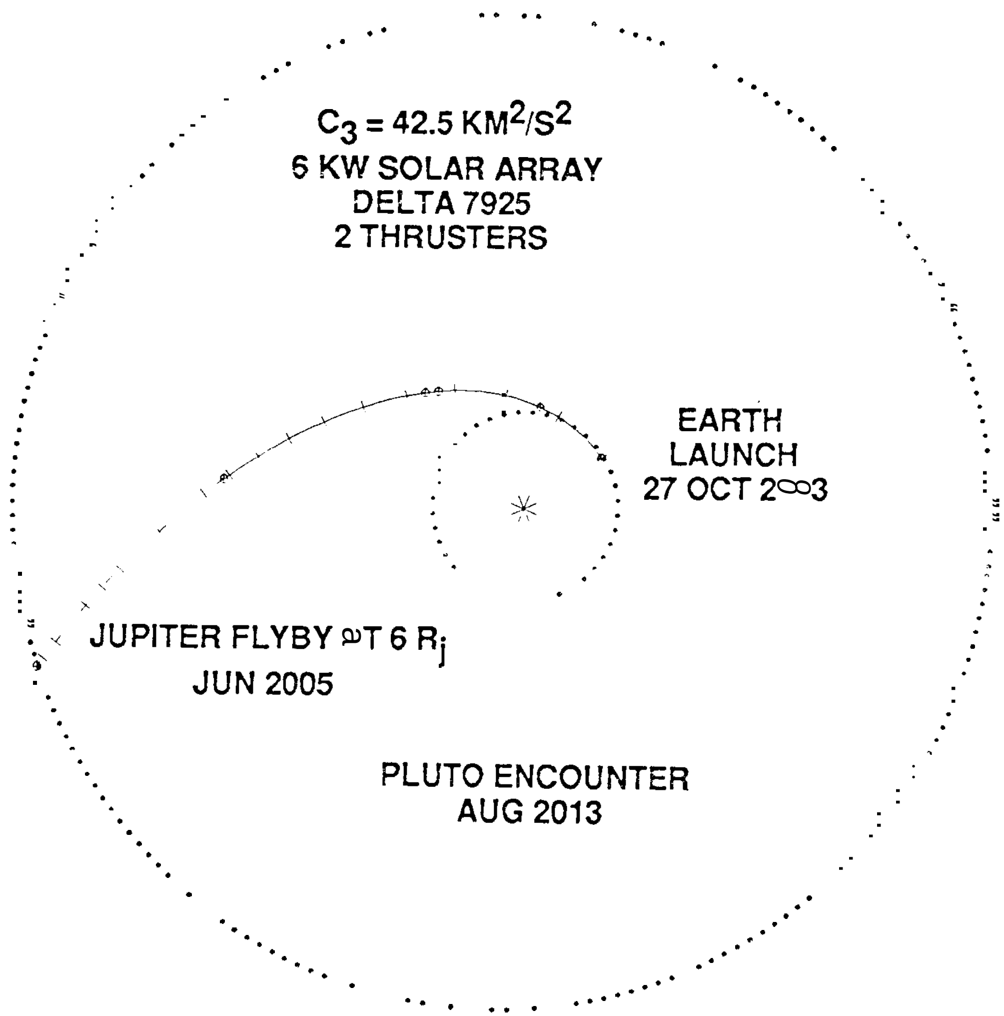
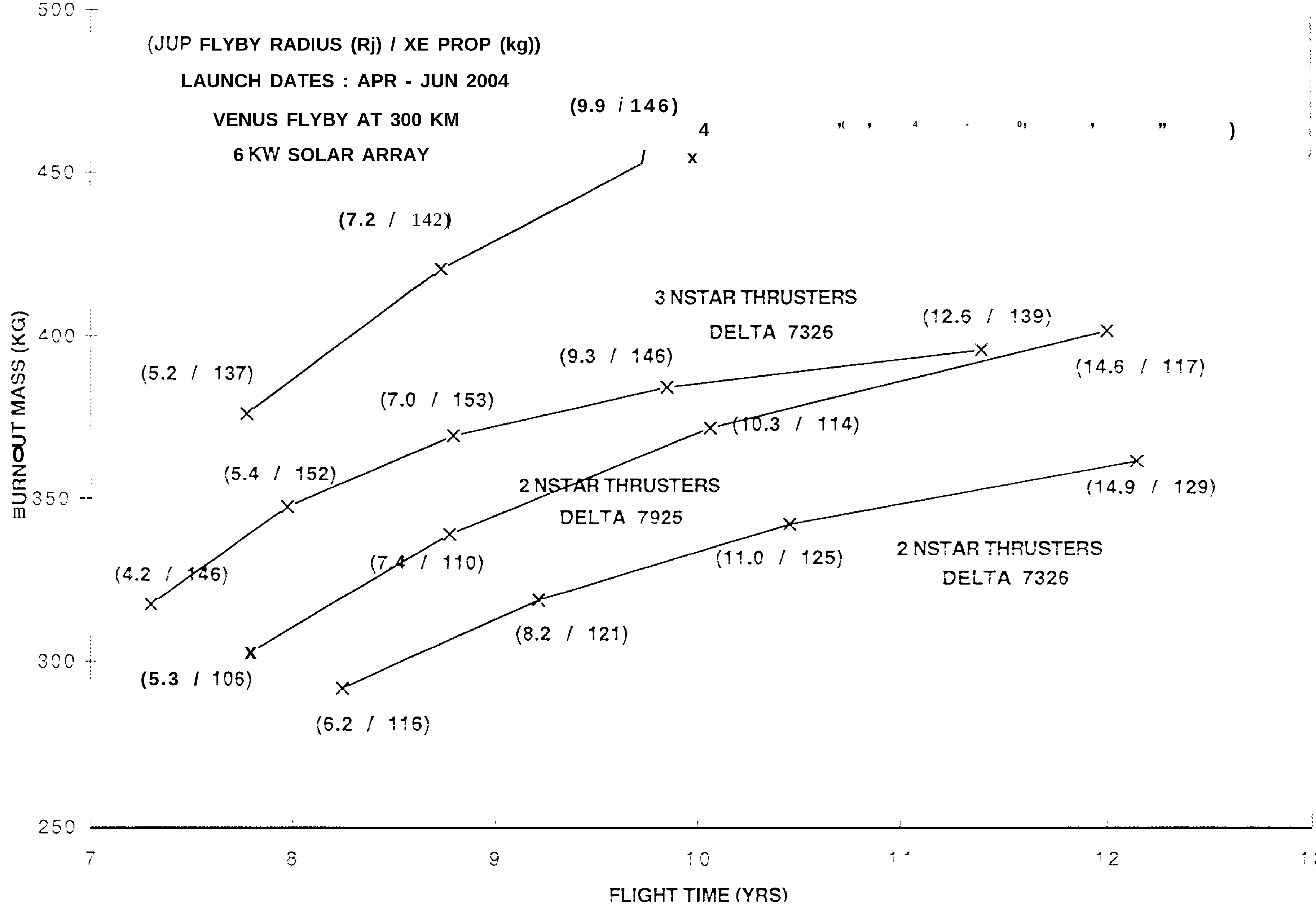

500
450
400
350
300
250
BURNOUT MASS (KG)
(JUP FLYBY RADIUS (Rj) / XE PROP (KG))
LAUNCH D᠁TES : OCT - NOV 2003
(2.4 / 62)
(3.2 / 65)
(4.4 / 67)
(6.0 / 70)
(8.4
11.8 / 74)
DELTA 7925
6 KW SOLAR ARRAY
(5.0 / 102)
(6.7 / 106)
(9.2 / 109)
(12.8 / 112)
3 NSTAR THRUSTERS
DELTA 7326
8 KW SOLAR ARRAY
7
8
9
10
1
12
13
FLIGHT TIME (YRS)

FIG 5



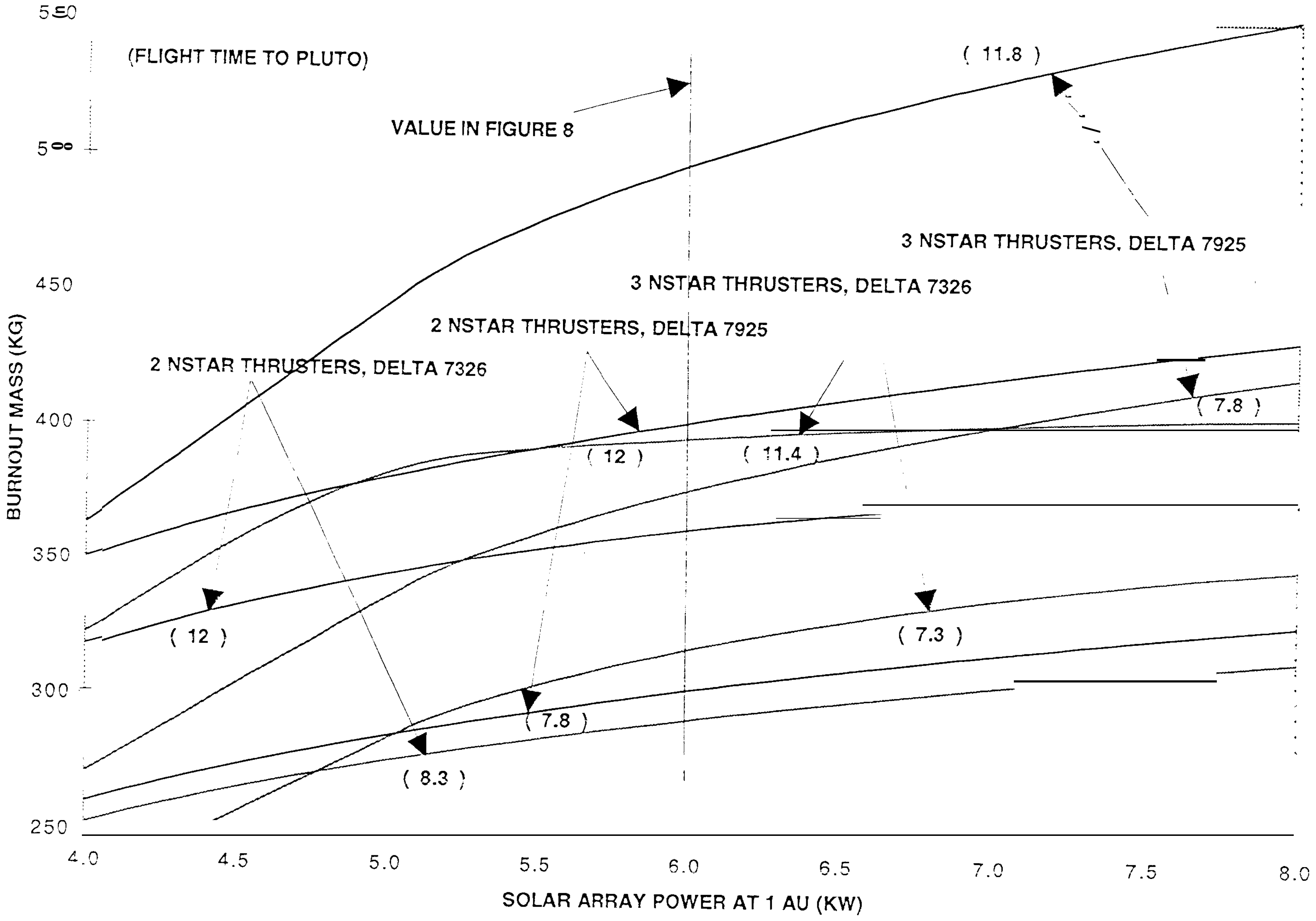
2 NSTAR THRUSTERS
DELTA 7925
6 KW SOLAR ARRAY
VALUE IN FIGURE 5
12.6 YEARS TO PLUTO
9.8 YEARS TO PLUTO
BURNOUT MASS (KG)
550
500
450
400
350
300
250
4.0
4.5
5.0
5.5
6.0
6.5
7.0
7.5
8.0
SOLAR ARRAY POWER AT 1 AU (KW)


$C_3 = 42.5\ KM^2/S^2$
6 KW SOLAR ARRAY
DELTA 7925
2 THRUSTERS
EARTH
LAUNCH
27 OCT 2003
JUPITER FLYBY AT 6 R_j
JUN 2005
PLUTO ENCOUNTER
AUG 2013



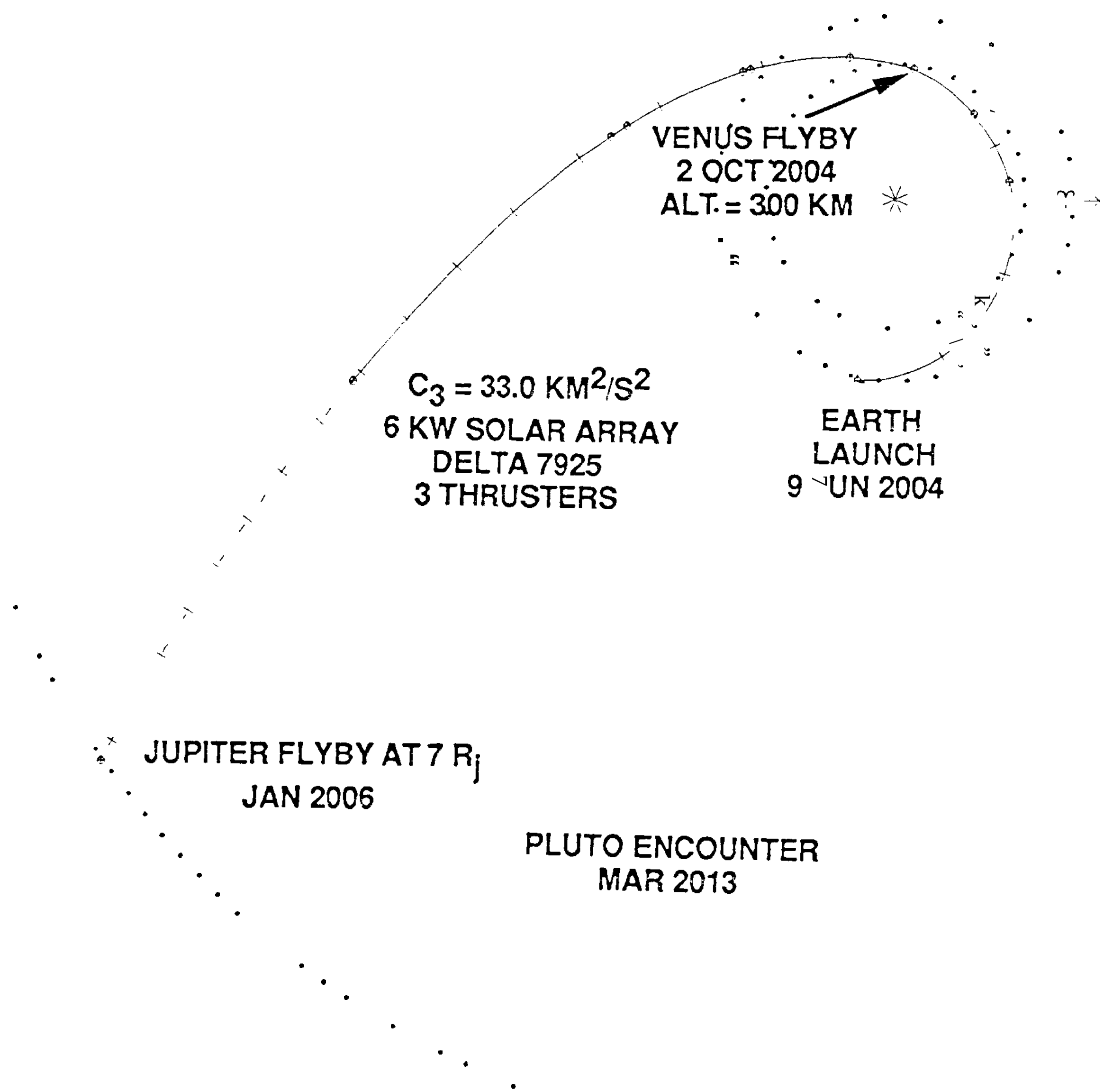
(JUP FLYBY RADIUS (Rj) / XE PROP (kg))
LAUNCH DATES : APR - JUN 2004
VENUS FLYBY AT 300 KM
6 KW SOLAR ARRAY
(9.9 / 146)
(7.2 / 142)
(5.2 / 137)
3 NSTAR THRUSTERS
DELTA 7326
(12.6 / 139)
(9.3 / 146)
(14.6 / 117)
(7.0 / 153)
(10.3 / 114)
(5.4 / 152)
2 NSTAR THRUSTERS
DELTA 7925
(14.9 / 129)
(4.2 / 146)
(7.4 / 110)
(11.0 / 125)
2 NSTAR THRUSTERS
DELTA 7326
(8.2 / 121)
(5.3 / 106)
(6.2 / 116)
BURNOUT MASS (KG)
500
450
400
350
300
250
7
8
9
10
11
12
13
FLIGHT TIME (YRS)

F 168



550
500
450
400
350
300
250
BURNOUT MASS (KG)
(FLIGHT TIME TO PLUTO)
VALUE IN FIGURE 8
(11.8)
3 NSTAR THRUSTERS, DELTA 7925
3 NSTAR THRUSTERS, DELTA 7326
2 NSTAR THRUSTERS, DELTA 7925
2 NSTAR THRUSTERS, DELTA 7326
(7.8)
(12)
(11.4)
(12)
(7.3)
(7.8)
(8.3)
4.0
4.5
5.0
5.5
6.0
6.5
7.0
7.5
8.0
SOLAR ARRAY POWER AT 1 AU (KW)

FIG 9



VENUS FLYBY
2 OCT 2004
ALT. = 300 KM
$C_3 = 33.0\ KM^2/S^2$
6 KW SOLAR ARRAY
DELTA 7925
3 THRUSTERS
EARTH
LAUNCH
9 JUN 2004
JUPITER FLYBY AT 7 R_j
JAN 2006
PLUTO ENCOUNTER
MAR 2013

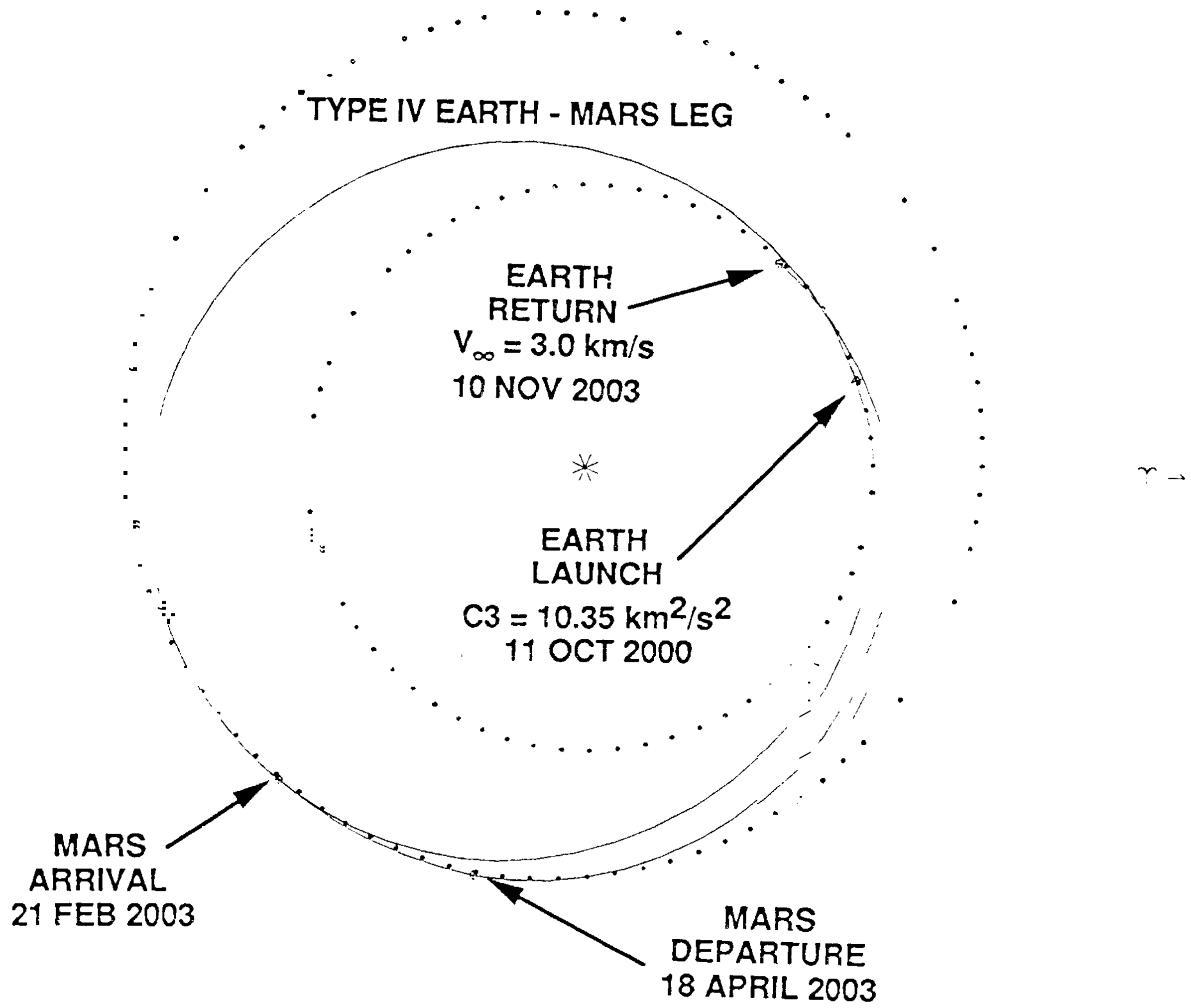

TYPE IV EARTH - MARS LEG
EARTH RETURN
V_∞ = 3.0 km/s
10 NOV 2003
EARTH LAUNCH
C3 = 10.35 km^2/s^2
11 OCT 2000
MARS ARRIVAL
21 FEB 2003
MARS DEPARTURE
18 APRIL 2003

Fig. 11

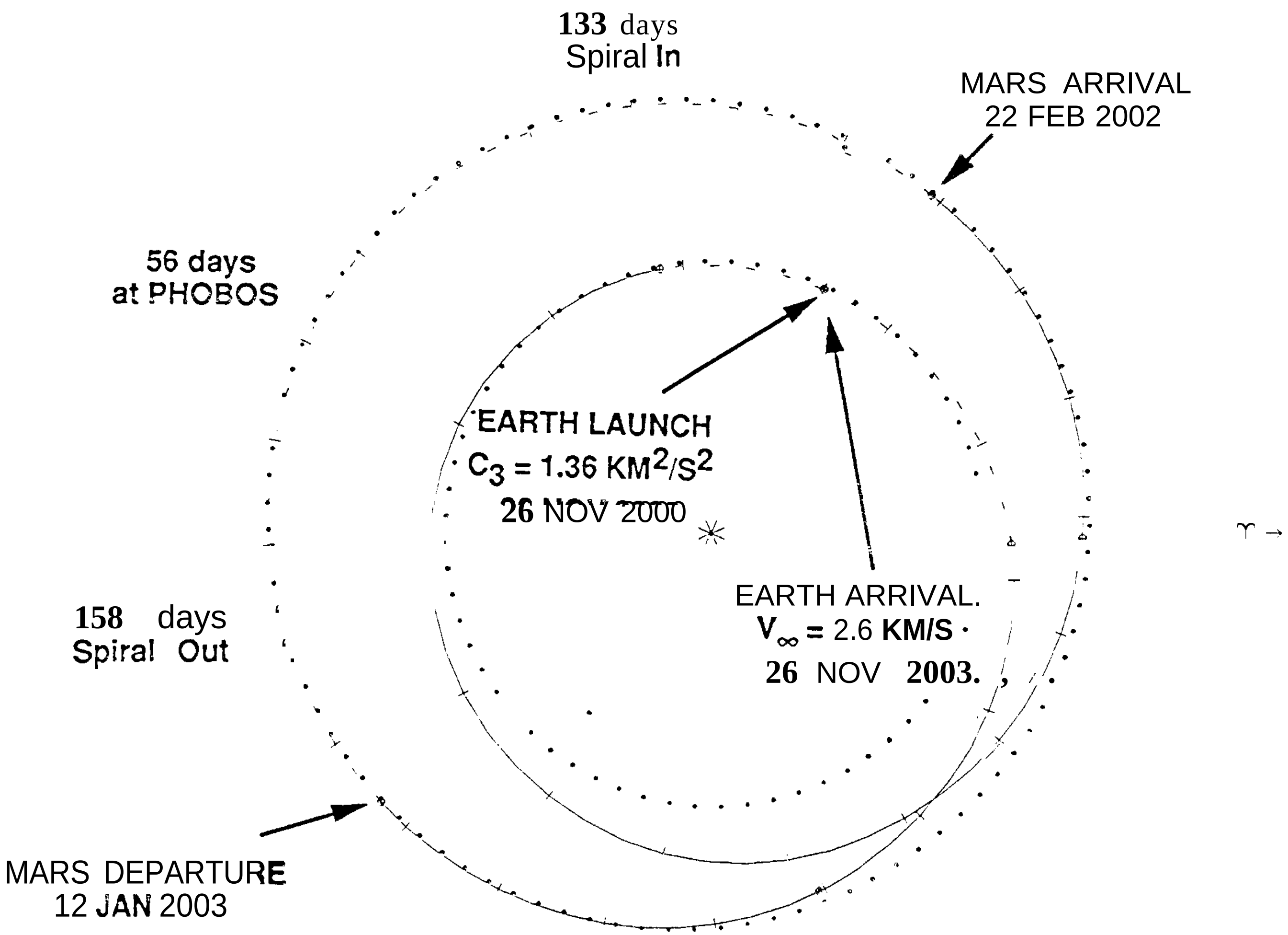

133 days
Spiral In
MARS ARRIVAL
22 FEB 2002
56 days
at PHOBOS
EARTH LAUNCH
C_3 = 1.36 KM^2/S^2
26 NOV 2000
EARTH ARRIVAL.
V_∞ = 2.6 KM/S
26 NOV 2003.
158 days
Spiral Out
MARS DEPARTURE
12 JAN 2003
♈ →

Fig 12